EO-1 ICD-019
Baseline
February 23, 1998

Earth Orbiter 1 (EO-1) Spacecraft to Pulsed Plasma Thruster (PPT) Interface Control Document

National Aeronautics and
Space Administration

Goddard Space Flight Center
Greenbelt, Maryland

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TBD List

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–	Initial Release		2/23/98
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Section 1. Scope

This interface control document (ICD) defines the interfaces between the Pulsed Plasma Thruster (PPT) and the Earth Orbiter-1 (EO-1) spacecraft, as well as the (flight and ground) functional, physical, environmental, and operating characteristics and other requirements related to meeting the objectives of the experiment.

This ICD will serve as the controlling technical document between the PPT and the EO-1 spacecraft. The document is controlled by the Goddard Space Flight Center (GSFC) EO-1 Project Office.

Section 2. Documents

The following documents of the exact issue shown form a part of the ICD to the extent specified herein. In the event of conflict between this ICD and the document referenced herein, the contents of this ICD shall be considered a superseding requirement.

2.1 Applicable Documents

SAI-PLAN-130	EO-1 Integration and Test Plan
SAI-PLAN-138	EO-1 Contamination Control Plan
SAI-SPEC-158	EO-1 Verification Plan and Environmental Specification
AM149-0020(155)	System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom
TBD	Mission Assurance Requirements
A0759	PPT-to-Spacecraft Interface Control Drawing
GSFC-426-EO-001	**Mission Assurance Document**

2.2 Reference Documents

GSFC-PPL	GSFC Preferred Parts List (latest issue)
MIL-M-38510	General Specification for Microcircuits
MIL-S-19500	General Specification for Semiconductors
MIL-STD-1547	Electronic Parts, Materials, and Processes for Space and Launch Vehicles
MIL-STD-975	Standard (EEE) Parts List
MIL-STD-202	Test Methods for Electronic and Electrical Components
MIL-STD-883	Test Methods and Procedures for Microelectronics
AM149-0030(155)	EO-1 Command Specification, Litton Amecom
AM149-0031(155)	EO-1 Telemetry Specification, Litton Amecom
TBD	PPT User's Manual

Section 3. Interface Requirements

3.1 Interface Definition

The PPT is a single-module, electromagnetic propulsion system, which uses Teflon as a propellant. For EO-1, one PPT module with two thrust-producing electrode/fuel bar assemblies will be mounted to the spacecraft with thrust vectors parallel to the spacecraft Z axis. The PPT will produce positive or negative pitch torque by selectively discharging through one of the two electrode pairs. The PPT experiment will use only the PPT to control in the spacecraft the pitch axis, torquer bar and the pitch momentum wheel pitch commanding will be disabled. The spacecraft will provide power, commands, mounting surface and fasteners, control software, harnessing, and interface electronics for the PPT. The PPT will provide telemetry to the spacecraft and a mounting interface to the spacecraft, and will incorporate a thermal control design to maintain the PPT temperature.

3.1.1 Interface Functions

The functions provided to the PPT by the spacecraft, and conversely, are delineated in the following subsections.

3.1.1.1 Spacecraft Interface Functions

The following major interface functions shall be provided by the spacecraft:

a. Provision of primary power from the 28 VDC power bus to the PPT

b. Provision of three discrete digital CMOS-driven TTL command lines from the spacecraft to the PPT

c. Provision of three analog telemetry lines to monitor voltages in the PPT

d. Provision of two analog telemetry lines to monitor PPT temperatures

e. Provision of two analog telemetry lines to monitor fuel gauges

f. Provision of mounting surface, inserts, thermal isolators, fasteners, and flight grounding strap

g. Provision of internal spacecraft harnesses for all PPT command and telemetry signals

h. Provision of mounting location for two external bulkhead connectors, which will electrically connect the PPT harnesses to the internal spacecraft harnesses

i. Provision of software necessary for PPT operation and experiment

3.1.1.2 PPT Interface Functions

The following major interface functions shall be provided by the PPT:

- a. Transmission of seven analog telemetry signals from the PPT to the spacecraft

b. Provision for mounting the PPT as defined in Interface Control Drawing A0759

c. Provision of harnesses for all PPT command and telemetry signals from the PPT unit to two bulkhead connectors on the external spacecraft surface

d. Provision of PPT and spacecraft bulkhead connectors, connector savers, and connector caps

e. Provision of break-out box for PPT-to-spacecraft bulkhead connection

f. Provision of nonflight electrode shorting plugs

3.2 Mechanical/Thermal Interface Requirements

The PPT experiment consists of a single unit. The PPT is mounted on the exterior of the spacecraft Bay 6 equipment panel. Threaded inserts shall be supplied by the spacecraft contractor, on the exterior of the panel, for mounting the PPT at the locations specified in Interface Control Drawing A0759. The PPT unit shall be configured such that removal from the spacecraft is possible after it is installed on the spacecraft.

3.2.1 Configuration

The configuration of the PPT on the Bay 6 equipment panel is shown in Figure 3-1.

3.2.1.1 Coordinate Systems

Orthogonal reference axes are established for the EO-1 spacecraft and the PPT. The PPT coordinate system is shown in Interface Control Drawing A0759. The EO-1 coordinate system is shown in Figure 3-2.

3.2.1.2 PPT Orientation

The PPT orientation is such that the spark plug # 1 electrode side is in the +Z spacecraft axis direction, as shown in Figure 3-3.

3.2.1.3 Fields of View

The PPT shall be located on the spacecraft such that a minimum 10-deg half-angle cone clear field of view (FOV) is maintained for both thrust nozzles. A 40-deg half-angle cone clear FOV is desired for each thrust nozzle, as shown in Figure 3-4.

3.2.1.4 Alignment With the Center of Mass

The PPT will be located on the spacecraft such that the Z component of the PPT thrust vectors are located within 20 cm in the spacecraft Y direction of the spacecraft beginning- and end-of-life center of mass.

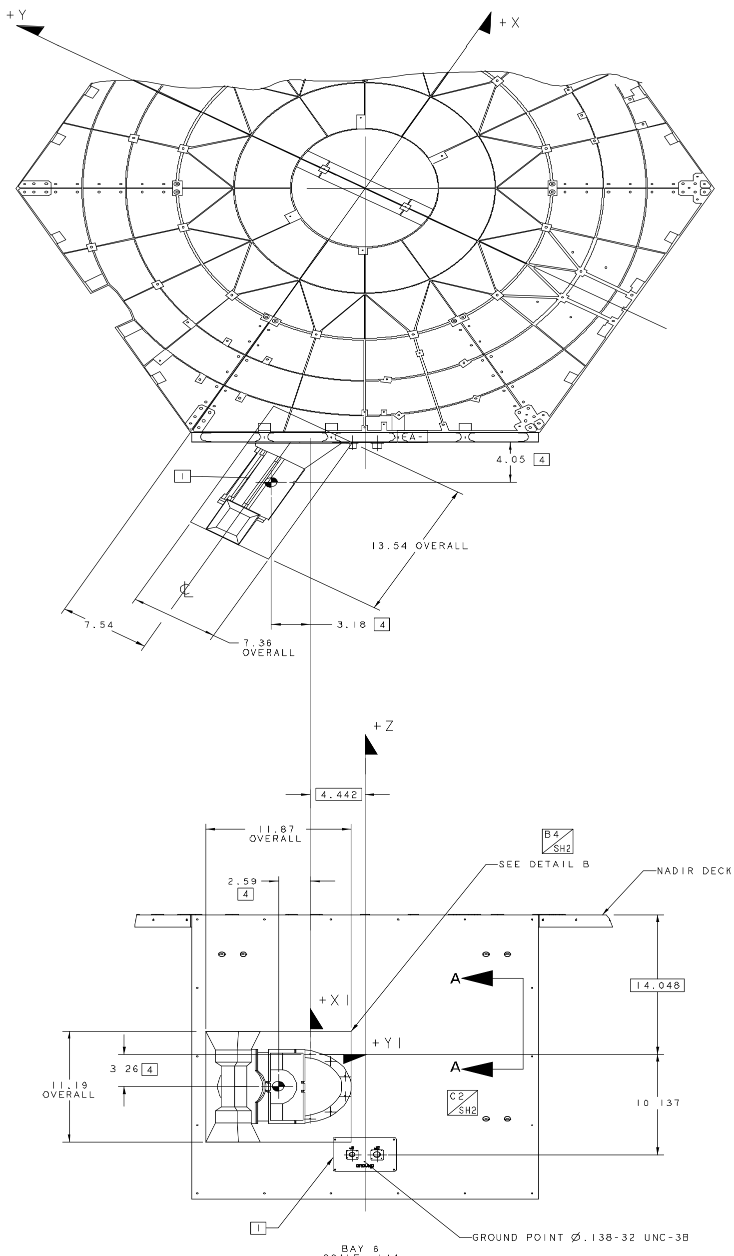


Figure 3-1. Configuration of the PPT (From Interface Control Drawing A0759)

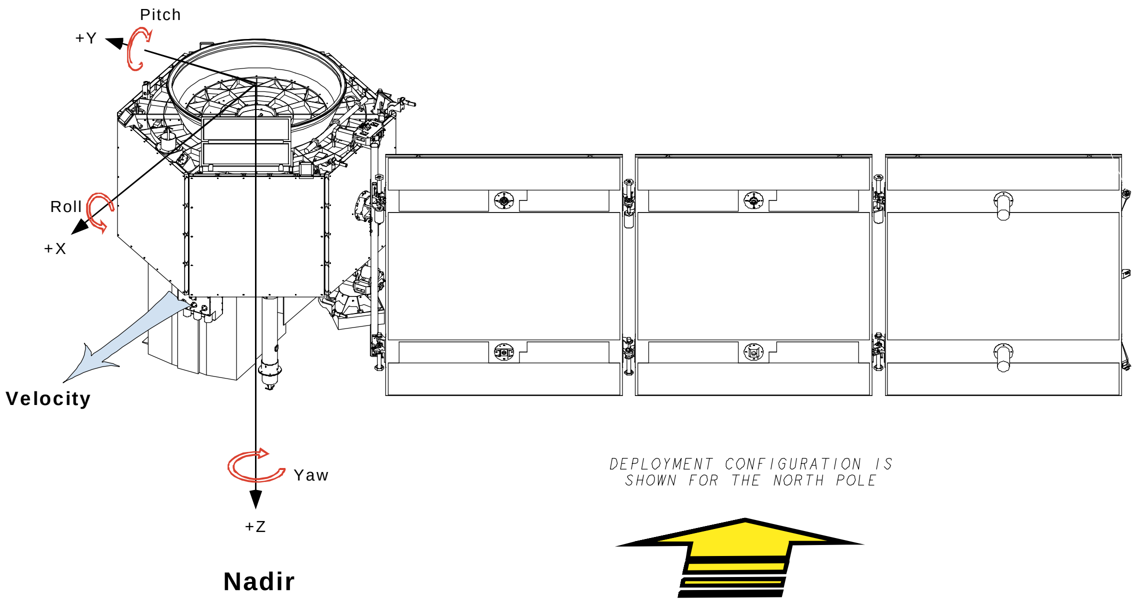


Figure 3-2. Deployed Spacecraft With Coordinate System (Sun Is Normal to the Page)

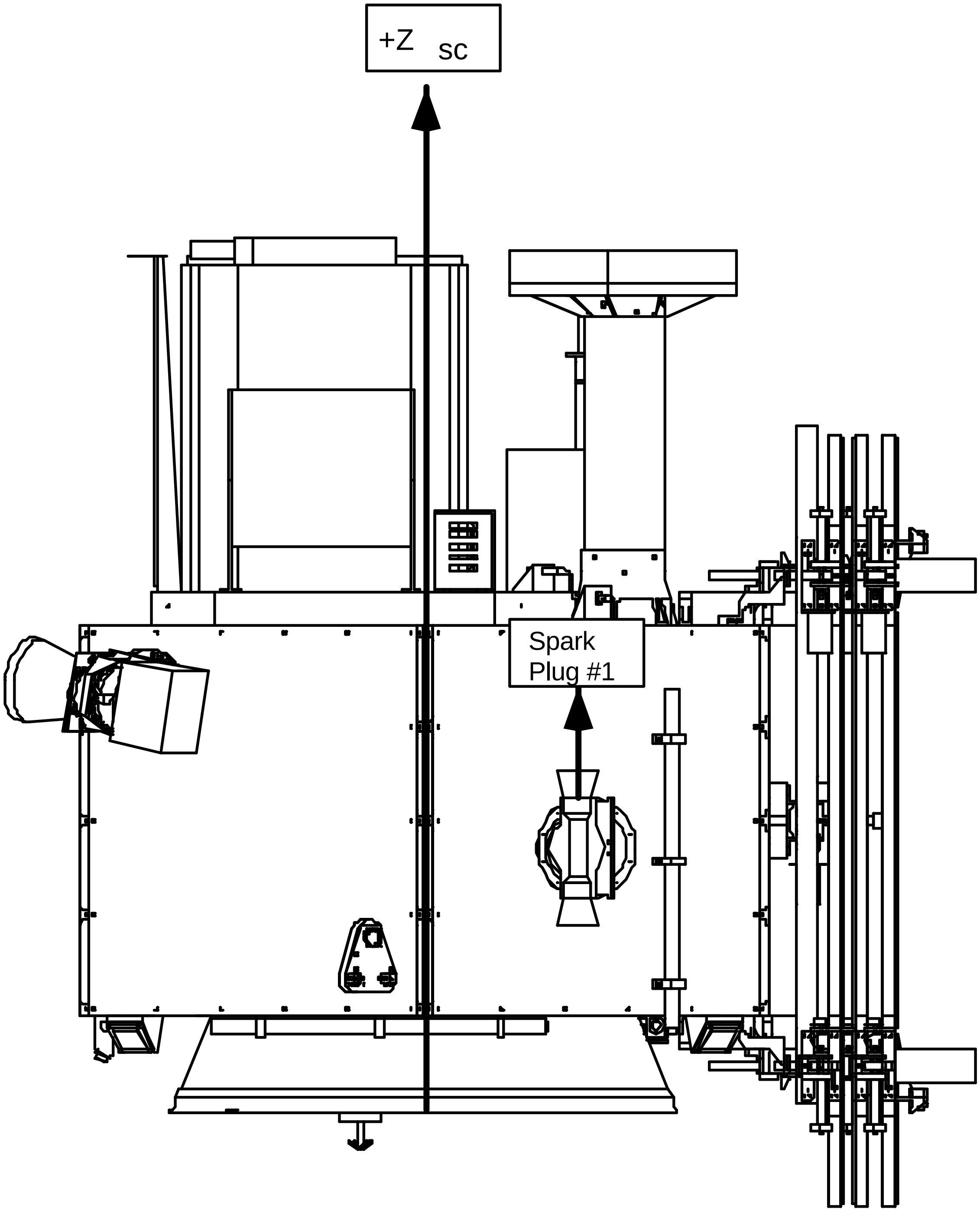


Figure 3-3. Electrode Orientation Drawing

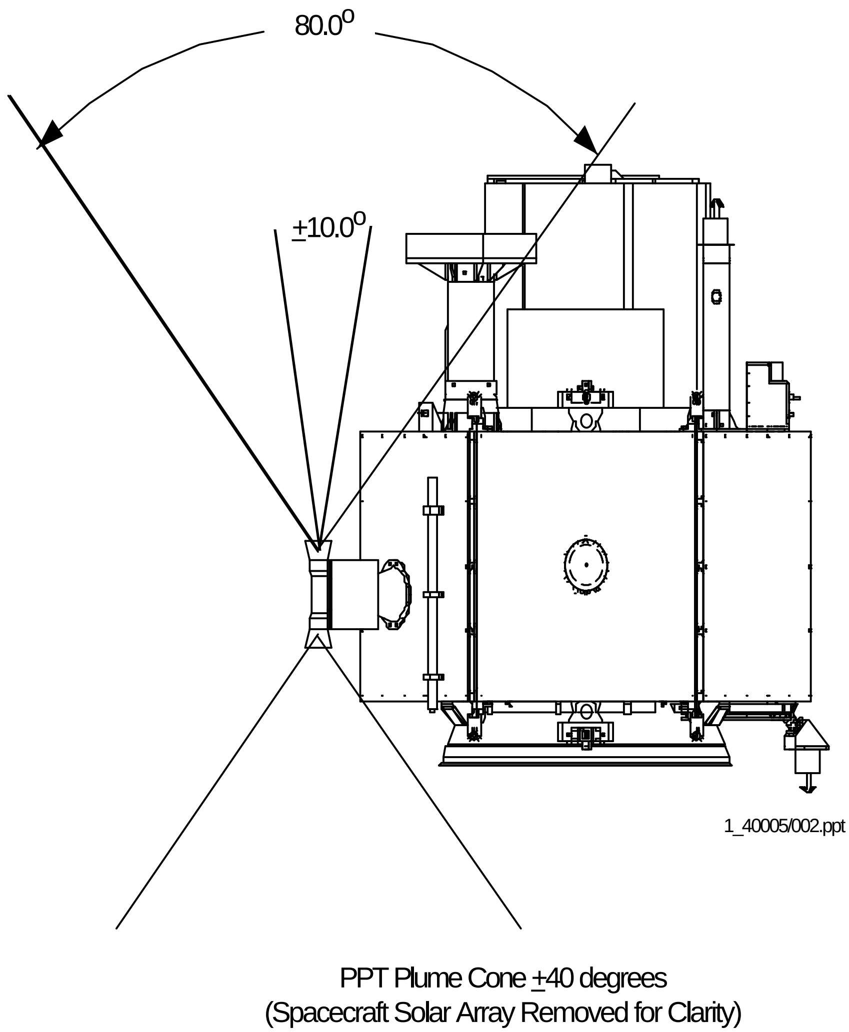


Figure 3-4. PPT Field-of-View Drawing

3.2.1.5 Mounting Interface

The PPT unit will mount directly to the spacecraft with 12 threaded fasteners and inserts to be supplied by the spacecraft. G10 thermal isolators will be used at the mounting surface and shall be provided by the spacecraft. The mounting pattern and thermal isolators are shown in Interface Control Drawing A0759.

3.2.1.5.1 Flatness Specification

Neither side of the mechanical interface plane shall be out of plane more than 0.25 mm.

3.2.1.5.2 In-Plane Accuracy

The mounting point centerlines shall not change more than 0.25 mm from nominal.

3.2.2 Mass Properties

Table 3-1 delineates the mass, dimensions, and center of gravity (CG) of the PPT unit.

Table 3-1. Mass Properties

Property	?????
Mass	NTE 6 kg
Dimensions	The dimensions of the PPT shall conform to Interface Control Drawing A0736.
Center of Gravity	The center of gravity of the PPT unit with respect to the c.g. location shown in Interface Control Drawing A0759 is ± 2.54 cm in each axis.

3.2.2.1 Mass

The total weight of the PPT shall not exceed 6 kg. All changes in mass estimates, including expected growth, shall be reported promptly. The final PPT mass shall be calculated to an accuracy of ±0.1 kg.

3.2.2.2 Center of Gravity

The final PPT CG shall be calculated to ±2.54 cm (1 inch).

3.2.2.3 Moment of Inertia

The moment of inertia (MOI) of the PPT about the PPT reference axis shall be calculated with 5 percent accuracy.

3.2.3 Mechanical Design and Analysis Requirements

All hardware shall be designed to survive the environments specified in the EO-1 Verification Plan and Environmental Specification, SAI-SPEC-158. All hardware shall be designed and analyzed to the applicable safety factors defined in Table 3-2. The analyses shall indicate a positive margin of safety. Limit loads are defined as the maximum expected flight loads.

Table 3-2. Material Factors

All Flight Hardware Except Pressure Vessels	Test Qual	Analysis Only
Material yield factors	1.25	2.0
Material ultimate factors	1.4	2.6

All ground support handling hardware shall have a design factor of safety of 5 (ultimate loads) and test to a minimum factor of safety of 2 without any permanent deformation occurring.

3.2.3.1 Limit Load Factors

The hardware shall be designed to withstand the quasi-static limit load (with applicable safety factors) defined in Table 3-3. This load should be applied in any direction at the component CG.

Table 3-3. Limit Load Factor

±15 g

3.2.3.2 Structural Stiffness Requirement

In the launch configuration, the PPT shall have a first mode frequency greater than 100 Hz and will verify this by analysis. A finite element model of the EO-1 satellite will be generated to be used in the launch vehicle coupled loads analysis. To aid in this effort, the mass properties of the deliverable hardware will be required.

3.2.3.3 Stress Analysis Requirement

A stress analysis shall be performed to verify the integrity of the component structure and attachments when subjected to the specified loads with the applicable safety factors. Margins of safety shall be determined, dominant failure modes identified, and this information transmitted to the satellite integrator. Existing mechanical stress analysis reports and data may be used if applicable.

3.2.3.4 Fastener Capacity

The deliverable hardware will be attached to the spacecraft panel using 12 threaded fasteners. A positive margin factor of safety shall be maintained for all the fasteners used on the spacecraft. The maximum load on any fastener shall not exceed 150 lb axial and 275 lb shear when subject to the quasi-static limit loads defined in Section 3.2.3.1.

3.2.3.5 Random Vibration

All hardware shall be designed to withstand the random vibration environment (with applicable safety factors) defined in Table 3-4.

3.2.4 Alignment

The total worst-case repeatable mechanical mounting alignment of the PPT with the spacecraft shall be less than 0.5 deg. No provisions shall be made for making alignment adjustments. The alignment of the surface of the fuel **faces** with respect to the holes will be measured to better than 0.5 deg.

3.2.5 PPT Handling Operations

The PPT User's Manual defines the handling and installation procedures for the PPT. The PPT will be installed by the spacecraft contractor with support from PPT personnel. Normal care shall be exercised during handling and installation of the equipment. Protective covers shall be supplied by the PPT contractor.

Table 3-4. PPT Random Vibration Test Levels

Frequency (Hz)	Level	
	Acceptance	**Protoflight**
20	0.006 g^2/Hz	0.011 g^2/Hz
20-100	+6 dB/octave	+6 dB/octave
100-500	0.14 g^2/Hz	0.28 g^2/Hz
500-2000	-6 dB/octave	-6 dB/octave
2000	0.009 g^2/Hz	0.018 g^2/Hz
Overall	10.64 grms	15.04 grms

NOTES:
1. Levels are for each of three orthogonal axes, one of which is normal to the mounting surface.
2. Levels are to be applied at the interface with the EO-1 spacecraft.
3. Test duration is 1 minute per axis.
4. The table shows flight acceptance and protoflight test levels. These levels may be reduced (notched) in specific frequency bands, with Project concurrence, if required to preclude damage resulting from unrealistic high amplification resonant response due to the shaker mechanical impedance and/or shaker/fixture resonances.
5. Flight-type attach hardware (including any thermal washers, etc.) shall be used to attach the test article to the test fixture, and preloads and fastener locking features shall be similar to the flight installation.
6. Cross-axis responses of the fixture shall be monitored during the test to preclude unrealistic levels.
7. During the test, the test article shall be operated in a mode representative of that during launch.

3.2.6 Access Requirements

Access requirements to the PPT shall be defined in the PPT User's Manual. Access requirements include connector mate/demate clearances, removal and replacement clearances for protective covers, and access to install and remove GSE required for safe discharge of the PPT.

3.2.7 Thermal

The specific types of thermal control available to the technology provider are radiation to space and regulated conductive paths to the spacecraft. Temperature control of the PPT will be accomplished by using selected thermal control coatings and multilayer insulating (MLI) blankets and by regulating the heat flow between the PPT and the spacecraft structure. The PPT and spacecraft panel temperature limits are defined in Section 3.2.7.1, and the spacecraft-provided thermal isolators are described in Section 3.2.7.2.

3.2.7.1 Thermal Interface

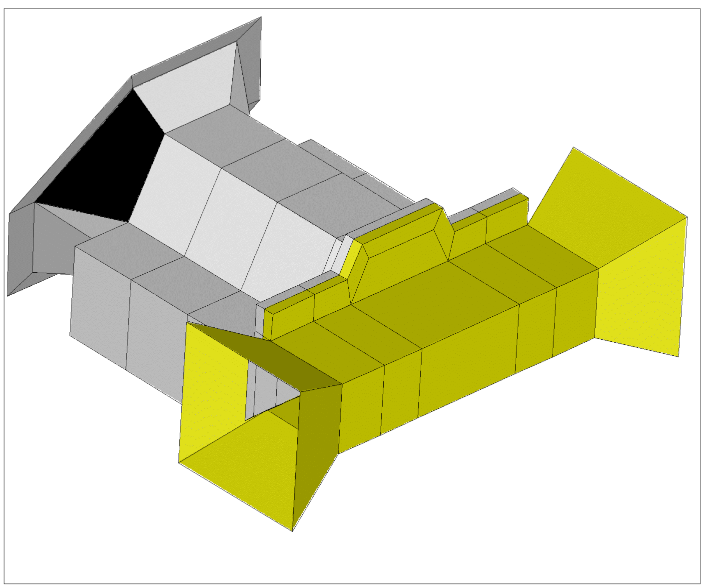
The thermal conductance between the spacecraft and the PPT shall not be greater than 0.22 W/C and the spacecraft interface temperatures shall not exceed the limits specified in Table 3-6. A monitorint point, defined in the PPT thermal data package, shall be within the limits specified in Table 3-6 A. The optical surface properties of the PPT are shown in Figure 3-5.

Table 3-6. Temperature Limits

Component	Operational Mode Limits	Survival Mode Limits
Spacecraft panel	0 to 40° C	- 10 to 50° C

Table 3-6.A PPT Thermal Monitoring Point Limits

	On-Orbit Survival	Thermal-Vacuum Test
Monitoring Point	**-22 to 30 C**	**-32 to 42 C**



White Polyurethane: e = 0.9+/- 0.05, absorptivity
Horn: e = 0.8, absorptivity

Figure 3-5. PPT Optical Surface Properties

3.2.7.2 Thermal Isolators

The spacecraft shall provide two thermal isolators for each mounting bolt. The inner thermal isolator between the spacecraft panel and the PPT flange is to have an O.D. of 0.6 inch and an I.D. of 0.25 inch with a thickness of 0.25 inch. The outer thermal isolator between the PPT flange and bolt head is to have an O.D. of 0.5 inch and an I.D. of 0.25 inch with a thickness of 0.25 inch. These isolators will be made out of G10.

3.2.7.3 Design Responsibility

The PPT vendor is responsible for the thermal design, thermal coatings application, and testing of the PPT. The spacecraft contractor is responsible for the thermal analysis of the combined PPT and spacecraft. The technology provider shall provide a thermal model of the PPT to the spacecraft contractor. The PPT-supplied thermal model shall include a maximum of 50 TRASYS surfaces and a maximum of 5 Sinda nodes.

3.3 Electrical Interface Requirements

The spacecraft will provide the power, command, and telemetry to operate the PPT by means of electronics located in the PSE and ACE. The PPT will provide two harnesses, one for power and one for command and telemetry, from the PPT to the spacecraft. Mating of the harnesses will take place at the spacecraft bulkhead connectors shown in Interface Control Drawing A0759.

The PPT provider shall provide electrical schematics for power input and current limit circuits, command interfaces, telemetry interfaces, and temperature sensors interfaces.

3.3.1 Power Requirements

3.3.1.1 Description

The PPT power system block diagram is shown in Figure 3-6.

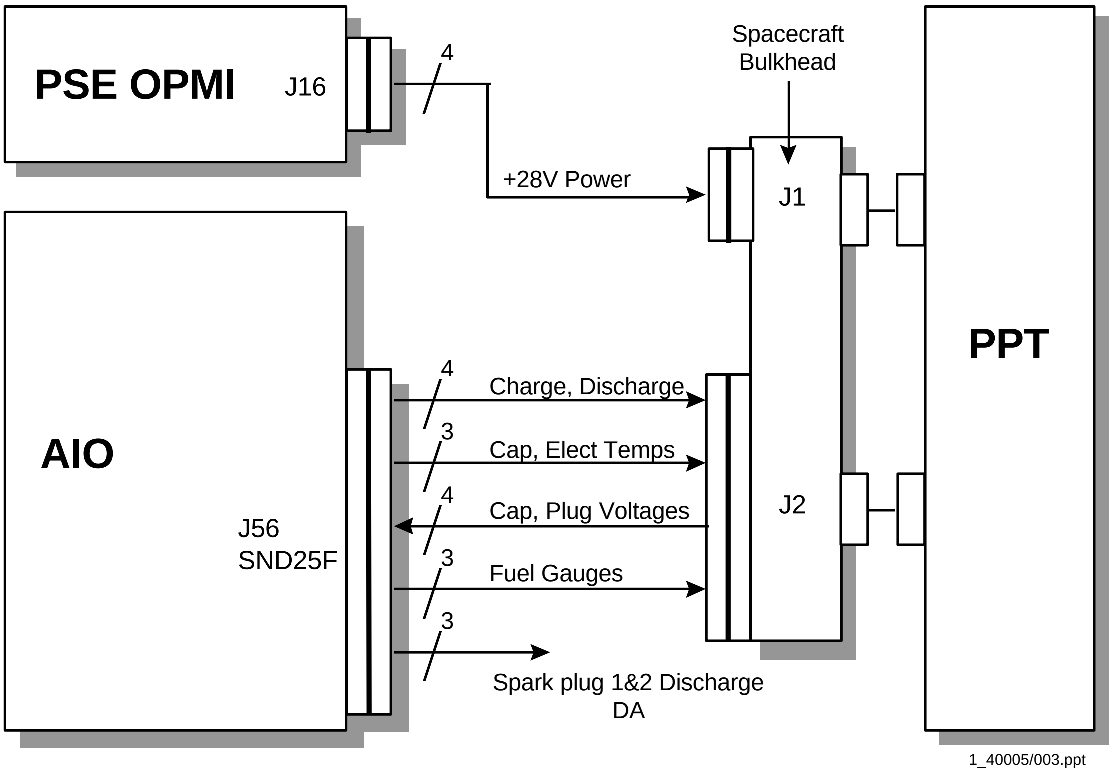


Figure 3-6. Electrical Functional Diagram of ACE/PPT Interface

3.3.1.2 Power Characteristics

The spacecraft will supply the PPT with the voltage and power characteristics listed in Table 3-7. The PPT provider shall ensure that the PPT shall operate successfully within this power regime.

Table 3-7. Power Requirements

Category	PPT Power
Voltage range	**28 ± 6 V**
Maximum current	**4 .5A**

3.3.1.2.1 Transients, Ripples and Spike Performance, Output Impedance

The power transients due to load switching and fault conditions, the ripple and spike performance of the supplied power, and the output impedance are defined in System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom document AM149-0020(155).

3.3.1.3 PPT Load Characteristics

The PPT's internal current-limiting circuit shall be designed such that the main capacitor is charged to **60** J when a 920-ms charge command is sent with a maximum input voltage of 34 V. The PPT shall be able to survive a voltage drop to zero for 20 sec. without damage to the PPT.

3.3.1.3.1 Power Distribution

The total PPT power allocation is given in Table 3-8. Nominal operation refers to the operation of the PPT for pitch attitude control during spacecraft nominal science model. Standby mode refers to mission phases in which the PPT is powered on but no commands are being sent to the PPT. Survival mode refers to all phases of the mission in which the PPT is not operated or in the standby mode.

Table 3-8. Power Allocation

Mode	Power
Nominal operation, orbit average	40 W
Standby mode	1 W

The current draw at nominal input voltage of 28 V is shown in Figure 3-7. The maximum current draw for any given input voltage is not greater than TBD A above nominal.

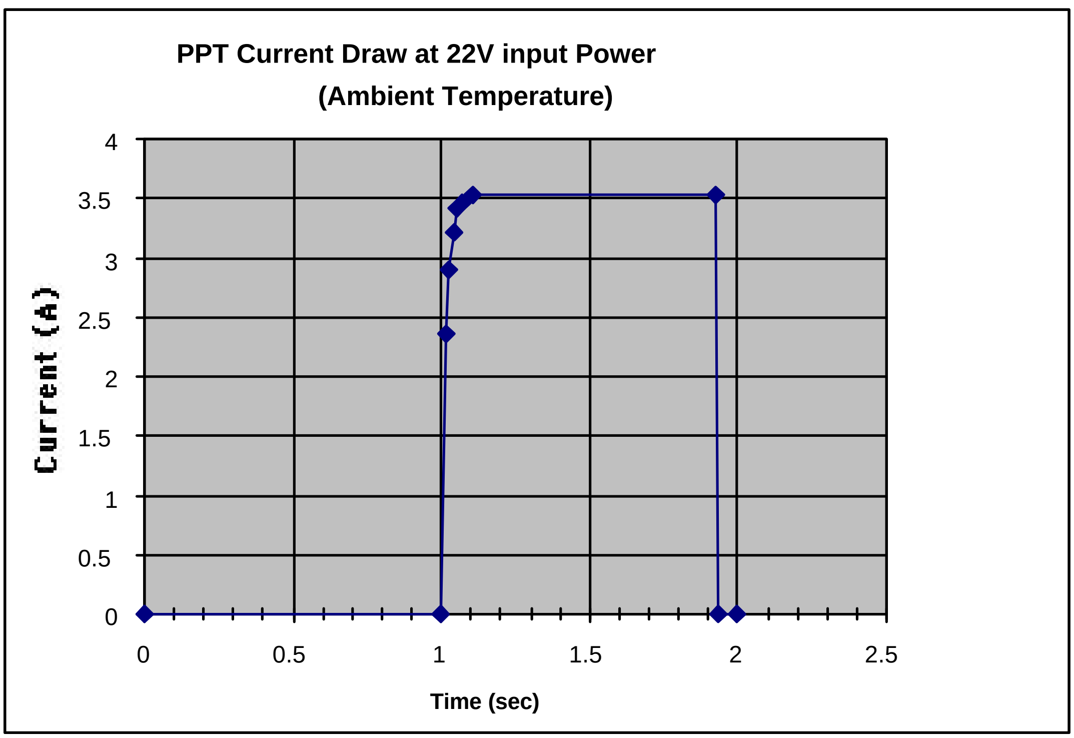


Figure 3-7. Charging Current Profile

3.3.1.3.2 Nominal Operation Load Profile

The nominal operation load profile of the PPT is illustrated in Figure 3-8.

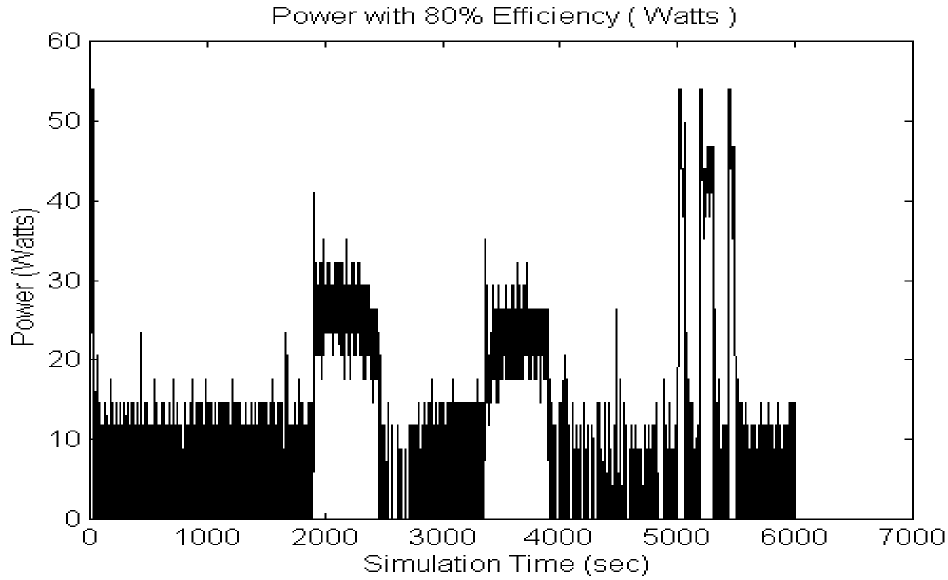


Figure 3-8A. Nominal Operation Load Profile

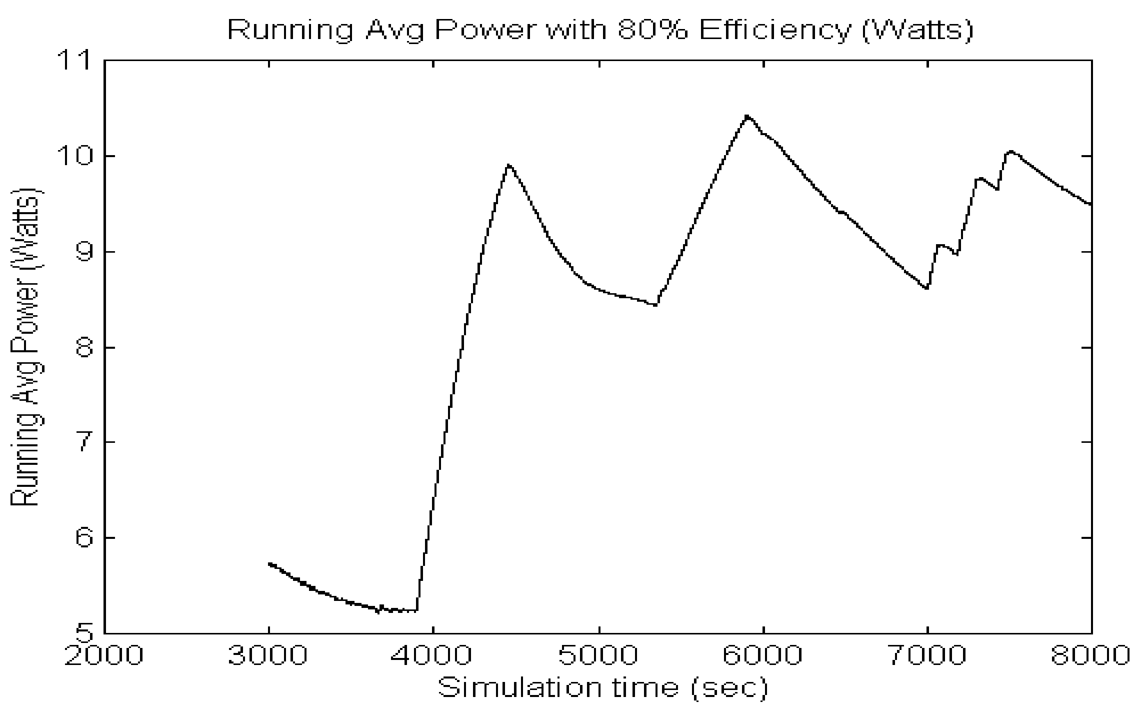


Figure 3-8B.

3.3.1.3.3 PPT Turn-On Transients

Refer to the PSE output turn-on transient definition in the System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom document AM149-0020(155).

3.3.1.3.4 Turn-Off Transients, Operational Transients, and Reflected Ripple and Spikes

Refer to System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom document AM149-0020(155).

3.3.2 Command Requirements

The **PPT** command **sequence for the PPT is illustrated** in **Figure** 3-9**. A description of each command and a description of the electrical characteristics of all commands are given below.**

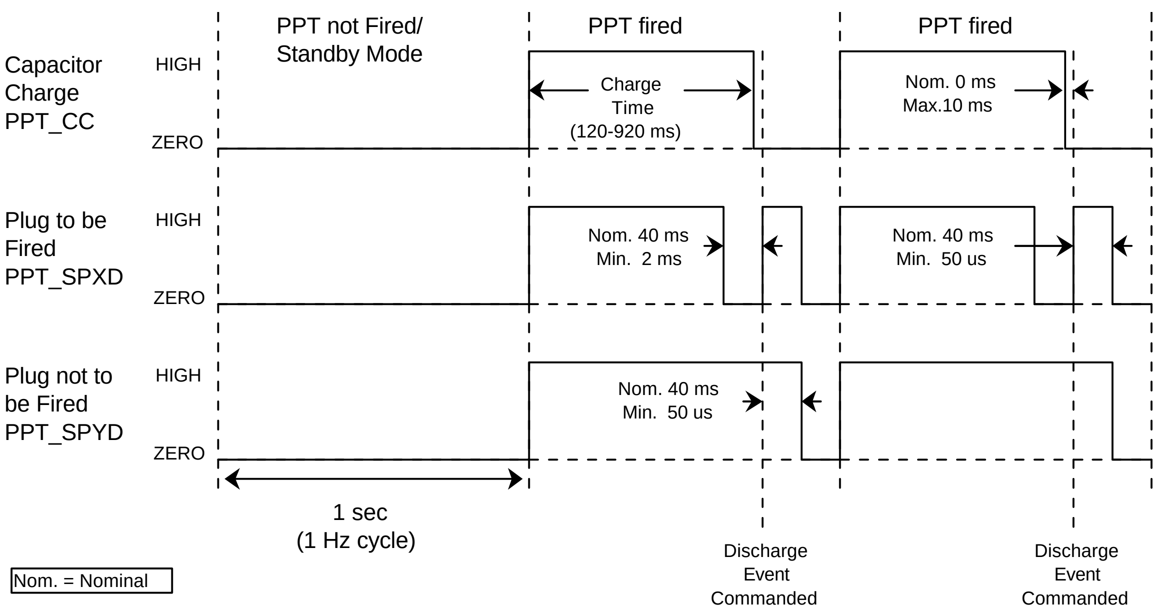


Figure 3.9 Command Pulse Sequence

3.3.2.1 Capacitor Charge

The PPT will charge the main capacitor for the length of time the capacitor charge **command** signal is switched from ~~a~~ logic zero to ~~the~~ logic ~~level~~ high ~~position.~~ **per the command sequence constraints shown in the upper trace of Figure 3-9.**

Table 3-9. Command Requirements

Command	Square Wave Signal	Duration	Max. Current
Capacitor charge	+5 V (-0.3 V to 1.8 V) low (3.5 to 10 V) high	100 to 920 ms	0.5 mA at 5 V
Spark plug 1 discharge	+5 V (-0.3 V to 1.8 V) low (3.5 to 10 V) high	≥ 10 µs	2.5 mA at 5 V
Spark plug 2 discharge	+5 V (-0.3 V to 1.8V) low (3.5 to 10V) high	≥ 10 µs	2.5 mA at 5 V

3.3.2.2 Spark Plug Discharge

A spark plug discharge **command** signal will discharge the **selected** spark plug and cause the PPT to fire **within 50 µsec of the command transition from logic zero (low) to logic high per the command sequence constraints shown in the middle trace of Figure 3.9. The spark plug discharge command for the side that is not to be fired will receive the command sequence per the constraints shown in the lower trace of Figure 3.9.**

3.3.2.3 Command Signal Electrical Characteristics

The open circuit voltage at the PPT interface of any of the three command lines with respect to the common command return shall not go below -1.5 V continuously.

The maximum open circuit voltage at the PPT interface of any of the three command lines with respect to the common command return shall not exceed 0.5 V for a logic "zero" command.

For a logic "high" command, the current supplied to any of the three PPT command circuits shall be at least 7 mA current at a nominal voltage of 1.4 V at the PPT interface between the command line and the common command return.

The current supplied to any of the three-PPT command circuits shall not exceed 30 mA for any voltage at the PPT interface.

3.3.3 Telemetry Requirements

Table 3-10 Telemetry Requirements

Parameter	Type of Signal	Voltage	Current
Capacitor Voltage	Analog voltage output from PPT with 1Kohm impedance	**0 – 4.5V**	5mA @ 5V
Spark plug 1 voltage	Analog voltage output from PPT with 1Kohm impedance	**0 – 6.0V**	5mA @ 5V
Spark plug 2 voltage	Analog voltage output from PPT with 1Kohm impedance	**0 – 6.0V**	5mA @ 5V
		Impedance	
Capacitor Temperature	Current source provided by spacecraft impedance as a function of temperature	**(YSI #44906)**	
Transformer temperature	Current source provided by spacecraft impedance as a function of temperature	**(YSI #44906)**	
Fuel gauge #1 voltage	Current source provided by spacecraft; impedance as a function of gauge position	**2.21 – 5.05 kOhm**	
Fuel gauge #2 voltage	Current source provided by spacecraft; impedance as a function of gauge position	**2.21 – 5.05 kOhm**	

3.3.3.1 Voltage Telemetry

The PPT will provide the spacecraft with 0- through 5-V analog signals, which are proportional to the actual voltage on the capacitor and spark plugs 1 and 2. The signals are low impedance outputs and are limited to 6.2 V by means of a zener diode.

3.3.3.2 Temperature Sensors

The temperature sensors will be YSI model #44906 and will be supplied by the spacecraft. The spacecraft also will supply a current source to the sensors and measure the voltage to determine temperature.

3.3.3.3 Fuel Gauges

The fuel gauges are floating variable impedance devices (5 kΩ - 10 kΩ). The spacecraft will supply a current source to the gauges and measure the voltage to determine the fuel usage.

3.3.4 Connectors, Pin Assignments, and Wiring List

3.3.4.1 Connectors

The connectors listed in Table 3-11 will be used with type M85049/17XXN03 EMI backshells.

Table 3-11. Connectors

Connector	Type
P1 PPT harness (power)	MS27484T10F35P
J1 spacecraft bulkhead (power)	MS27472T10F35S
P2 PPT harness (signal)	MS27484T12F35S
J2 spacecraft bulkhead (signal)	MS27472T12F35P

3.3.4.2 Connector Mounting

The spacecraft bulkhead connector locations are shown on Interface Control Drawing A0759. The bulkhead connectors shall be supplied by the PPT supplier.

3.3.4.3 Pin Assignment/Wiring List

Table 3-12 provides the pin assignment/wiring list.

3.3.5 Electromagnetic Compatibility

3.3.5.1 EMC Requirements

The following subsections address conducted and radiated emission and susceptibility levels. These requirements, which make up the core of the EMC specification, are drawn from MIL-STD-461C.

Table 3-12. Pin Assignment/Wiring List

Description	Signal Name	Signal Type	Source Brd-Conn-Pin	Destination Brd-Conn-Pin	AWG	Notes
+28 V power #1	PPT-28A	Pwr,+28 V	PSEOM #1J56-1	PPT J1-1	22	2
+28 V power #1 return	PPT_28A_RTN	Pwr, Return	PSEOM #1J56-4	PPT J1-6	22	
+28 V power #2	PPT-28B	Pwr,+28 V	PSEOM #1J56-2	PPT J1-2	22	
+28 V power #2 return	PPT_28B_RTN	Pwr, Return	PSEOM #1J56-5	PPT J1-7	22	
Capacitor charge	PPT_CC	Dig, I bit	PIO J85-20	PPT J2-6	22	3
Spark plug 1 discharge	PPT_SP1D	Dig, I bit	PIO J85-8	PPT J2-1	22	3
Spark plug 2 discharge	PPT_SP2D	Dig, I bit	PIO J85-9	PPT J2-8	22	3
CCharge/Pdischarge return	PPT_CCPD_RTN	Dig, Return	PIO J85-21	PPT J2-5	22	
Capacitor temperature	PPT_TEMPC	Thermister	PIO J85-4	PPT J2-12	22	4
Transformer temperature	PPT_TEMPT	Thermister	PIO J85-5	PPT J2-10	22	4
PPT temperature return	PPT_TEMP_RTN	Ana, Return	PIO J85-17	PPT J2-11	22	
Capacitor voltage	PPT_CAPV	Ana, 0 to +5V	PPT J2-4	PIO J85-23	22	5
Spark plug #1 voltage	PPT_SP1V	Ana, 0 to +5V	PPT J2-2	PIO J85-10	22	5
Spark plug #2 voltage	PPT_SP2V	Ana, 0 to +5V	PPT J2-7	PIO J85-11	22	5
Spark cap/pug 1&2 return	PPT_CPV_RTN	Ana, Return	PPT J2-3	PIO J85-24	22	
Fuel gauge 1 voltage	PPT_FG1	5-10 k ohms	PIO J85-12	PPT J2-15	22	6
Fuel gauge 2 voltage	PPT_FG1	5-10 k ohms	PIO J85-13	PPT J2-13	22	6
Fuel 1&2 voltage return	PPT_FGV_RTN	5-10 k ohms	PIO J85-25	PPT J2-14	22	

NOTES:
1. Thermostat control is inside the PPT.
2. Power wires are sized for 5A continuous through 22 AWG pairs.
3. Digital command lines from PIO card to PPT share a common digital return line.
4. Thermisters inside PPT have common return for current source on PIO card.
5. Analog voltage telemetry lines from PPT to PIO card share a common analog return line.
6. Potentiometers inside PPT share common return for current source on PIO card.

3.3.5.1.1 Conducted Emissions

The unit shall comply with the conducted emissions requirements found in Section 3.2.8 of System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom document AM149-0020(155).

3.3.5.1.2 Conducted Susceptibility

The unit shall not exhibit any malfunction, degradation of performance, or deviation from specified indications beyond the tolerances specified herein when subjected to the following electromagnetic energy signals injected onto its dc power leads:

Ripple	2.8 V RMS or 40 W at any frequency from 30 Hz to 50 kHz, 1 V RMS or 1 W at any frequency from 50 kHz to 400 MHz
Transients	+28 or -28 V, zero-to-peak, 10 µs width, at any repetition rate up to 300 Hz (50 Ω source)

3.3.5.1.3 Radiated Emissions

The unit shall comply with the radiated emissions requirements found in Section 3.2.8 of System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom document AM149-0020(155).

3.3.5.1.4 Radiated Susceptibility

The unit shall not exhibit any malfunction, degradation of performance, or deviation from specified indications beyond tolerances specified herein when subjected to the radiated susceptibility requirements discussed in the following subsections.

3.3.5.1.4.1 Electric Field

The limits are as tabulated:

Frequency Range	Field Intensity (V/m)
14 kHz to 2 GHz	2
2 to 3 GHz	20
3.6 to 8.6 GHz	2
8.6 to 9 GHz	50
9 to 18 GHz	2

3.3.5.1.4.2 Magnetic Field

The magnetic field intensity shall be consistent with nearby magnetic torquer bar activity in the 30 to 60 Am^2 range, within 0.25 m.

3.3.5.2 Shielding

For harness shielding, shielding methods, and shielding termination and grounding requirements, refer to the System Level Electrical Requirements NMP EO-1 Flight, Litton Amecom document AM149-0020(155).

3.3.5.3 Isolation

The main discharge capacitor and the electrode/stripline assemblies returns will be electrically isolated from the spacecraft with at least ~~300~~ **1500 Ohm +/- 1 2% DC.** ~~Ohm impedance~~

3.3.6 Mechanical

The electrical requirements for mechanical bonding, bonding measurements, and chassis design shall conform to those requirements specified in the System Level Electrical Requirements NMP +EO-1 Flight, Litton Amecom document AM149-0020(155).

3.4 Environmental Requirements

The PPT shall be designed to survive and operate in the environments found in the EO-1 Verification Plan and Environmental Specification, SAI-SPEC-158.

3.4.1 Radiation

The PPT shall be designed to operate within specification in the environment specified in the EO-1 Mission Ionizing Radiation Specification, Attachment A.

3.4.2 Safety

The PPT presents only the following safety hazard:

- ONLY IF 28 V power is applied AND a command signal is given to charge the capacitor, potentially lethal voltages can be present on the electrodes, which are recessed in the horn assemblies and not easily accessible. Care must be exercised during periods in ground test during which 28 V is applied to the PPT.

3.4.3 Contamination

The PPT shall be fabricated and maintained in accordance with the contamination requirements specified in the EO-1 Contamination Control Plan, SAI-STD-138.

3.5 Software Interfaces

The spacecraft software will accommodate the command and telemetry requirements listed that are necessary to operate the PPT experiment.

3.5.1 Telemetry Requirements

Table 3-13 lists the telemetry requirements.

Table 3-13. Telemetry Requirements

	Size (bits)	Rate (Hz)	Source	Engineering Units	Measured Range	Note
Parameters from PPT						
Capacitor voltage	10	5	PIO	0-1809 Volts	0 – 4.5 V	1
Spark plug #1 voltage	10	1	PIO	0-954 Volts	0 - 6.0V	2
Spark plug #2 voltage	10	1	PIO	0-954 Volts	0 – 6.0 V	2
Capacitor temperature	10	1	PIO	-45 – 65 ^{o}C	Non-linear	3
Transformer temperature	10	1	PIO	-45 -100 ^{o}C	Non-linear	3
Fuel gauge #1	10	1	PIO	0 – 25.0 mm	2210 – 5050 Ohm	4
Fuel gauge #2	10	1	PIO	0 – 25.0 mm	2210-5050____	4
Parameters from s/c______________						
PPT power status	1	1	PIO	On/Off	----	----
PIO voltage	10	1	PIO	volts	----	----
Required PPT pulse	10	1	ACS	#1/#2 sec	----	----
Commanded PPT charge time	10	1	ACS	#1/#2 sec	----	----
Discharge spark plug #1 command	10	1	ACS	Fire/nofire	----	----
Discharge spark plug #2 command	10	1	ACS	Fire/nofire	----	----
Total # of spark plug #1 discharges	10	1	ACS	counts	----	----
Total # of spark plug #2 discharges	10	1	ACS	counts	----	----
Cumulative charge time	10	2	ACS	sec	----	----

Notes

1: The spacecraft will capture the capacitor voltage during the charge interval immediately prior to the fire command and immediately after the discharge command

2: The spacecraft will capture the spark plug voltage during the charge interval immediately prior to the fire command

3: GSFC s-311-P-18, Thermistor model #44906

4: Range is slightly beyond limit for fuel gauge, but PPT has additional fuel beyond the limit for fuel gauge.

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3.5.2 Ground Commands

The spacecraft will provide the capability of receiving the following ground commands:

Command	Destination	Comments
Power PPT ON	**0 - 920 msec**	
Power PPT OFF	**0 - 920 msec**	
Charge capacitor for xxx sec. and discharge spark plug #1	PIO	xxx is restricted to values between (TBD-TBD)
Charge capacitor for xxx sec. and discharge spark plug #2	PIO	xxx is restricted to values between (TBD-TBD)
Enable PPT control mode	ACS	
Disable PPT control mode	ACS	

3.5.3 ACS

The EO-1 flight ACS software will incorporate the logic and associated processing functions to allow the PPT be used as a pitch attitude control device.

Section 4. GSE

The PPT provider will supply the following GSE:

a. A device for connecting to electrodes to allow for safe discharge of PPT in ambient conditions and protect the electrodes during integration and test (I&T)

b. Handling fixtures/transportation box and electrical break-out boxes if required

c. Electrical break-out box to connect each of the two PPT connectors to the spacecraft bulkhead connectors

d. Hardware to enable the safe discharge of th PPT in ambient conditions when integrated to the spacecraft.

e. Shorting plug for the PPT to prevent inadvertent charging of the PPT.

f. Electrical breakout boxes to connect to each of the two PPT harnesses.

g. A contamination control bag which will allow for the discharge of the PPT during spacecraft thermal vacuum testing while preventing condensable contamination from being exhausted into the vacuum chamber.

The spacecraft integrator will provide electrical GSE capable of commanding and performing command and data handling (C&DH) functional tests of the PPT during I&T.

Section 5. Deliverables List

The PPT supplier will provide the following items to the spacecraft vendor:

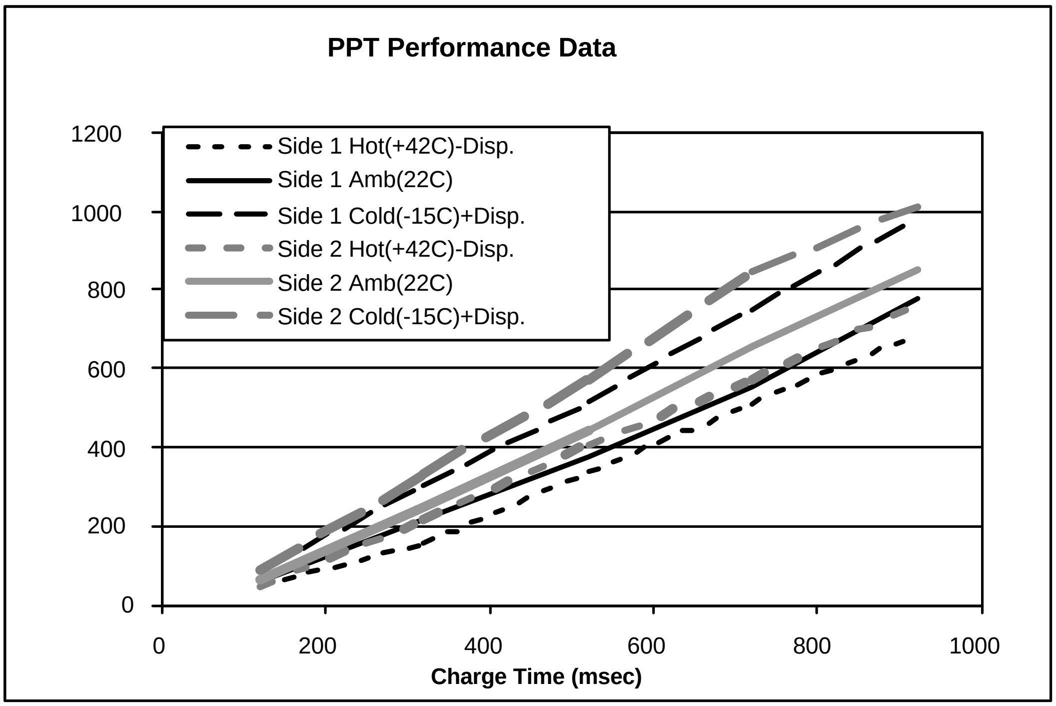
Item No.	Item	Deliver Date
1	Flight PPT unit	4 Sept 99
2	GSE to enable safe charge and discharge of PPT while command through s/c	4 Sept 99
3	Safety Shorting plug for electrodes	4 Sept
4	Each of the two spacecraft bulkhead connector with backshells	15 June 99
5	A break-out box for each of the two PPT connectors	4 Sept
6	Connector savers for the PPT and spacecraft bulkhead connectors	4 Sept
7	Connector caps for the PPT and spacecraft bulkhead connectors	4 Sept
8	Handling/transportation fixture for PPT	4 Sept
9	PPT User's Manual with safety analysis	
10	Acceptance Test Data Package	10 Sept
11	Mechanical Analysis Package	4 Sept
12	Thermal Model and Analysis Package	4 Sept
13	PPT Functional Test Procedure and PPT Integration and Test Procedure	14 Sept

Section 6. PPT Performance

The PPT shall be designed to deliver the following performance on orbit.

6.1 Impulse Bits

The nominal impulse bit produced **by each of the two** PPT **electrodes** as a function of commanded charge time for an input voltage of 28 V is given in ~~Figure 6-1.~~ **the graph below**. The impulse bit varies with temperature for a given charge time. (Note: The temperatures shown are not on-orbit predictions of high, low, and nominal) The added dispersions to the high and low cases shown are half the peak to peak shot-to-shot variations. The graphs are constructed from the data listed in the following table.



charge time	Side 1 Hot (+42C)-Disp.	Side 1 Amb (22C)	Side 1 Cold (-15C)+Disp.	Side 2 Hot (+42C)-Disp.	Side 2 Amb (22C)	Side 2 Cold (-15C)+Disp.
120	50	59	85	50	65	90
320	160	218	305	220	251	335
520	340	374	520	410	448	575
720	515	558	755	575	657	845
920	685	778	980	765	859	1015

6.2 Thrust Vector

The thrust vector of the PPT shall be within **5.5 deg**. of the geometric center of the horn assemblies.

6.3 Operational Constraints

The command charge time of the PPT will not be greater than 920 ms and will not be less than **160** ms. The PPT will not be command to fire more than once every 1-Hz cycle.

6.4 Operational Capability

Under nominal operations, defined by spacecraft attitude control system analyses, the PPT fuel mass shall provide a minimum of 30 days of continuous on-orbit operational control.

Abbreviations and Acronyms

μs	microsecond
°C	degree Celcius
Ω	ohm
A	ampere
ACE	?attitude control electronics?
ACS	
C&DH	command and data handling
CG	center of gravity
cm	centimeter
CMOS	
dB/octave	
deg.	degree
EMC	?electromagnetic compatibility?
EO-1	Earth Orbiter-1
FOV	field of view
g^2/Hz	
GHz	gigahertz
grms	
GSE	?ground support equipment?
GSFC	Goddard Space Flight Center
Hz	hertz
I&T	integration and test
I.D.	inside diameter
ICD	interface control document
J	joule
kΩ	kilohm
kg	kilogram
kHz	kilohertz

lb	pound
m	meter
MΩ	megohm
mA	milliampere
MHz	megahertz
MLI	multilayer insulating
mm	millimeter
MOI	moment of inertia
ms	millisecond
ns	nanosecond
NTE	
O.D.	?outside diameter?
PIO	
PPT	Pulsed Plasma Thruster
PSE	?power switching electronics?
RMS	?root mean square?
sec.	second
TBD	to be determined
TTL	
V	volt
V/m	volt per meter
VDC	
W	watt